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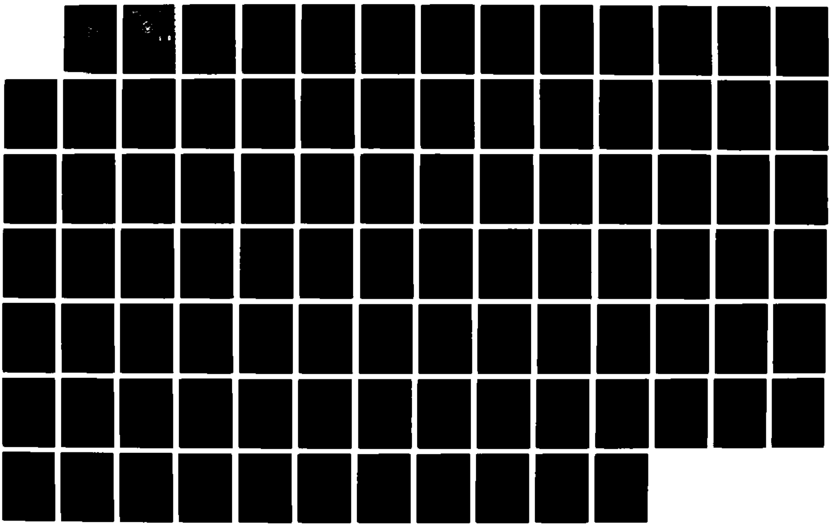

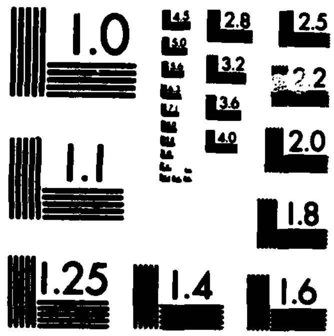

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Monterey, California

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THESIS

MICROCOMPUTER SOFTWARE SUPPORT
FOR CLASSES IN AIRCRAFT
CONCEPTUAL DESIGN

by

Michael Lee Cramer

March 1987

Thesis Advisor G. H. Lindsey

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87

UNCLASSIFIED
SECURITY CLASSIFICATION OF THIS PAGE

AD-A181 450

REPORT DOCUMENTATION PAGE

1a REPORT SECURITY CLASSIFICATION UNCLASSIFIED	1b RESTRICTIVE MARKINGS
2a SECURITY CLASSIFICATION AUTHORITY 2b DECLASSIFICATION / DOWNGRADING SCHEDULE	3 DISTRIBUTION / AVAILABILITY OF REPORT Approved for public release; distribution is unlimited.
4 PERFORMING ORGANIZATION REPORT NUMBER(S)	5 MONITORING ORGANIZATION REPORT NUMBER(S)
6a NAME OF PERFORMING ORGANIZATION Naval Postgraduate School 6b OFFICE SYMBOL (If applicable) 67	7a NAME OF MONITORING ORGANIZATION Naval Postgraduate School
6c ADDRESS (City, State, and ZIP Code) Monterey, California 93943-5000	7b ADDRESS (City, State, and ZIP Code) Monterey, California 93943-5000
8a NAME OF FUNDING / SPONSORING ORGANIZATION 8b OFFICE SYMBOL (If applicable)	9 PROCUREMENT INSTRUMENT IDENTIFICATION NUMBER
8c ADDRESS (City, State, and ZIP Code)	10 SOURCE OF FUNDING NUMBERS PROGRAM ELEMENT NO / PROJECT NO / TASK NO / WORK UNIT ACCESSION NO

11 TITLE (Include Security Classification)
MICROCOMPUTER SOFTWARE SUPPORT FOR CLASSES IN AIRCRAFT CONCEPTUAL DESIGN

12 PERSONAL AUTHOR(S)
Cramer, Michael L

13a TYPE OF REPORT	13b TIME COVERED	14 DATE OF REPORT (Year Month Day)	15 PAGE COUNT
Master's Thesis	FROM ______ TO ______	1987 March	92

16 SUPPLEMENTARY NOTATION

17 COSATI CODES			18 SUBJECT TERMS (Continue on reverse if necessary and identify by block number)
FIELD	GROUP	SUB-GROUP	Aircraft Conceptual Design Design, Conceptual Design, Aircraft

19 ABSTRACT (Continue on reverse if necessary and identify by block number)

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This thesis develops a computer program to enhance learning about design by performing calculations for aircraft conceptual design which follow hand calculation methods. It is intended to be used in the aircraft design course taught by the Department of Aeronautics at the Naval Postgraduate School, Monterey California.

20 DISTRIBUTION / AVAILABILITY OF ABSTRACT ☒ UNCLASSIFIED/UNLIMITED ☐ SAME AS RPT ☐ DTIC USERS	21 ABSTRACT SECURITY CLASSIFICATION UNCLASSIFIED
22a NAME OF RESPONSIBLE INDIVIDUAL G. H. LINDSEY	22b TELEPHONE (Include Area Code) (408) 646-2391 22c OFFICE SYMBOL Code 014

DD FORM 1473, 84 MAR
83 APR edition may be used until exhausted
All other editions are obsolete

SECURITY CLASSIFICATION OF THIS PAGE
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Microcomputer Software Support for Classes in Aircraft Conceptual Design

by

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Lieutenant Commander, United States Navy
B.B.A., University of Notre Dame, 1975

Submitted in partial fulfillment of the
requirements for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

from the

NAVAL POSTGRADUATE SCHOOL
March 1987

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ABSTRACT

The conceptual phase of aircraft design determines the general size and configuration of an aircraft. Many calculations are performed in assessing the optimum parameters. The calculations are often lengthy and iterative in nature and are thus highly appropriate for computer programing.

This thesis develops a computer program to enhance learning about design by performing calculations for aircraft conceptual design which follow hand calculation methods. It is intended to be used in the aircraft design course taught by the Department of Aeronautics at the Naval Postgraduate School, Monterey California.

TABLE OF CONTENTS

ACKNOWLEDGEMENTS

A special thanks is due my thesis advisor, Dean G. H. Lindsey, without whom this program would not have been possible. My thanks are also due to Lt. Bob Drake who provided the inspiration for this project with his thesis on Helicopter Design.

I. INTRODUCTION

Aircraft design is a graduate level course taught by the Department of Aeronautics at the Naval Postgraduate School, Monterey California. During this twelve week course, the student is required to perform numerous calculations, many of which are repetitive, in the evolution of a conceptual design of a fighter/attack aircraft. The iterative nature of aircraft design makes this task well suited to computer assistance; however, particular care must be exercized not to compromise the learning process by "over-automating" the process.

The objective of this thesis is to provide students with a tool that will enhance learning from the design experience during the limited course time available. This is achieved by eliminating some of the tedious manual calculations, particularly in the iterative procedures. The program was designed to be used on a personal micro-computer in view of their convenience and wide-spread availability. Every attempt has been made to display to the student the logic sequence involved in the program. In this respect the computer code has been optimized for learning. The same theory is employed in the software that students are using for their hand calculations, and intermediate results are

displayed to prevent the creation of a magic "black box", which would have little educational value.

Finally, it is hoped that this program will provide the framework for further additions and improvements. In this respect it is envisioned to be the first of several such programs, which will be incorporated into all aspects of the aircraft design course.

II. PROGRAM DESCRIPTION

The computer program written for this thesis is divided into ten chapters. ~~These chapters~~ which are addressed through a commom menu called the Chapter Selection Program; (Fig 2.1).

```
**** CHAPTER SELECTION PROGRAM ****
***********************************

            CHAPTERS
            ********

1) Introduction,

2) Preliminary Estimate of Take-off Weight,
3) Meeting Performance Requirements,
4) Aspect Ratio Optimization,
5) Wing Geometry Design,
6) Estimating Fuselage Length,
7) Tail Design,
8) Determining Structural Weights (WS),
9) Refined  Estimate of WTO Using WS, and

10) End Session ,   Theses),
```

Fig 2.1 Chapter Selection Program

The program is completely interactive and proceeds in stages which parallel the developments in the design course. The flow logic of the program is given in Fig. 2.2. Results of each calculation are displayed on the screen and summarized at the end of individual sections. For efficient operation, input and output data is stored in data files,

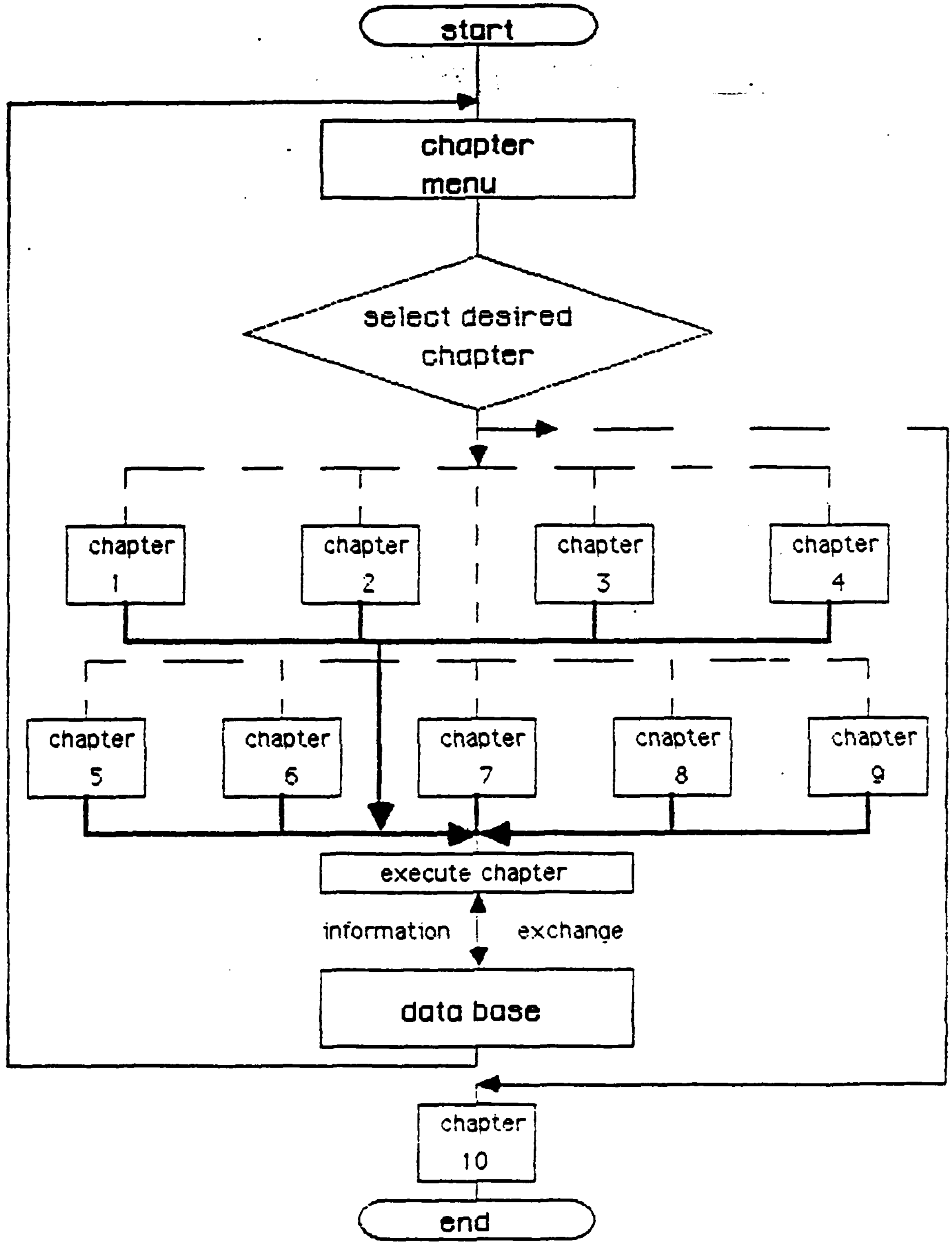

Figure 2.2 Computer Flow Logic

which are written onto the diskette to provide a common data base between chapters and to provide permanent storage of completed work. A single diskette is used for both the program and the data files for convenience of operation.

Each Chapter subject is discussed in detail during the Aircraft Design Course. The program is intended to supplement the course as a tool to expedite completion of a significant portion of the many calculations required. It is expected that by using this program the student will be able to progress more quickly through the material, while learning as much as before about it and still freeing time to cover additional topics.

III. MISSION AND PERFORMANCE REQUIREMENTS

A. PRELIMINARY ESTIMATES OF TAKE-OFF WEIGHT

1. Discussion

The design process begins with an estimate of take off weight, WTO. WTO is a very important design parameter because it sizes the entire vehicle. Since only the mission requirements are known initially, many assumptions must be made to get started. The characteristics and descriptive parameters of current aircraft, along with existing engines, are used in formulating the assumptions employed in the initial estimate of the required WTO.

Starting from a preliminary guess for WTO, the first refinement in its value can be made with a technique which employs final weight over inital weight fractions calculated for each phase of the mission. These fractions are found by using both empirical and theoretical relationships, which require as inputs the historical parameters from existing airplanes. In chapter five of Fundamental of Aircraft Design [1:5-1 - 5-24] Nicolai presents a method that uses seven phases to describe any mission profile. The fuel weight is determined by subtracting final weight from WTO, and the ratio of empty weight to take-off weight can be found from the following equations:

$$WTO = WF + WE + WPL \qquad (3\text{-}1)$$

where WF = fuel weight
WE = empty weight
WPL = payload weight.

The resulting relationship of empty weight as a function of take-off weight is then solved simultaneously using an historical regression line of WE versus WTO. The following section describes each of the seven phases as outlined by Nicolai and the calculations for WTO. Chapter two of the design program is an automation of this procedure.

2. Mission Profile Phases

a. Phase 1 - Engine Start and Take-off

The weight fraction for this phase is based on empirical data. Typical values are between .97 and .975.

$$\frac{W2}{WTO} \cong .9725 \qquad (3\text{-}2)$$

b. Phase 2 - Accelerate to Cruise Mach and Altitude

This fraction is derived from the outbound cruise mach. There exists an empirical relation between initial cruise mach and initial cruise altitude. Essentially, aircraft with higher cruise machs cruise at higher altitudes and use a larger percentage of their weight to complete the initial acceleration and climb phase. Nicolai demonstrates

this relationship graphically, and an excellent fit of the curve for subsonic cruise was obtained with the following linear relation:

$$W3/W2 = 1.0065 - (0.0325)\,(M1) \qquad (3\text{-}3)$$

where M1 is the outbound cruise mach.

c. Phase 3 - Cruise Out

The weight fraction for this phase is based on the Brequet range equation. The expression for a jet aircraft is

$$\frac{W4}{W3} = \exp\left(\frac{(-R)\ (c)}{(V)\ (L/D)}\right) \qquad (3\text{-}4)$$

where R = range
c = specific fuel consumption
V = velocity
L/D = lift/drag.

The optimum cruise velocity will maximize the ratio of W4/W3. This optimum is achieved by flying at a Mach number which is associated with a value of approximately 0.943 L/D_{max}. For modern high bypass engines, however, the variation of specific fuel consumption with mach is considerable and must be taken into account in the exact solution for optimum cruise Mach number.

d. Phase 4 - Acceleration to High Speed

The weight fraction for acceleration from a cruise condition to a high speed dash can be estimated with the following factors:

$$A1 = 1.0065 - (0.0325)(M1) \qquad (3\text{-}5)$$

where A1 is the weight fraction produced by acceleration from M = .1 to the cruise Mach number

$$A2 = 0.990 - (0.008)(M2) - (0.1)(M2^2) \qquad (3\text{-}6)$$

where A2 is the weight fraction produced by acceleration from M = .1 to the high speed dash Mach number

$$WLS = A1 / WI \qquad (3\text{-}7)$$

$$WHS = A2 / WI \qquad (3\text{-}8)$$

where WLS = Weight after accelerating from M = .1 to low speed
WHS = Weight after accelerating from M = .1 to high speed
WI = Weight at M = .1

Thus, the weight fraction after acceleration from cruise to high speed dash is:

$$W5/W4 = A2 / A1 \qquad (3\text{-}9)$$

e. Phase 5 - Combat

The fuel used during this phase is determined by the mission requirement for combat time and thrust level. Engine performance data must also be known.

$$\text{Combat fuel} = (c)\ (\text{thrust})\ (\text{time}) \tag{3-10}$$

where c is thrust specific fuel consumption.

Additional weight and drag changes occur if ordnance is dropped during this phase. The weight at the end of combat, W6, may then be expressed as:

$$W6 = W5 - \text{combat fuel} - \text{ordnance dropped} \tag{3-11}$$

f. Phase 6 - Cruise Back

The cruise back weight fraction is determined in the same manner as the cruise out fraction, substituting any changes in profile specifications as required.

$$\frac{W7}{W6} = \frac{(-R)\ (C)}{(V)\ (L/D)} \tag{3-12}$$

g. Phase 7 - Loiter

The loiter weight fraction may be determined by the classical equation as follows:

$$\frac{W8}{W7} = \exp\left(\frac{(-E)\ (C)}{(L/D)}\right) \tag{3-13}$$

where E is the endurance time and L/D is typically L/D_{max}.

3. Determining WTO

WTO is the sum of payload, fuel weight, and empty weight as shown in equation (3-1). The payload (ordnance and crew) is obtained from the mission specifications. The fuel weight is determined as a fraction of WTO from the calculations described in the previous section. The final relationship needed to solve for take-off weight is provided by a regression line of WE vs WTO based on historical trends for the type of aircraft being ananlyzed. The regression line relationship demonstrates the decreasing ratio of empty weight, WE, to WTO as WTO increases. This decrease in WE as a fraction of WTO occurs because the weight of many internal components is fixed; hence, the weight of the empty structure does not increase proportionately to WTO as weight increase.

If all of the mission weight changes were expressed in terms of weight fractions, the solution for WTO could be obtained directly. Unfortunately, the ordnance weight and combat fuel weight are fixed values, not weight fractions.

Because of these fixed values, the solution for WTO becomes an iterative process and, hence, well suited for a computer solution.

4. Sensitivity Studies

Additional advantages accrue from the computer solution in performimg sensitivity studies. These analyses allow the user to quickly change a single variable and quickly see the net effect on WTO. For example, the user would complete the analysis for a particular profile and then change a parameter such as ordnance load by a given amount. The resulting increase in WTO may be quite dramatic if the aircraft is sensitve to this parameter. One might typically find that for a one pound increase in ordnance carried, the take-off weight may increase four or five pounds. This occurs because of a multiplying effect whereby changing one requirement changes many others. The additional ordnance increases drag and adds weight. This in turn requires a stronger wing, which in itself adds weight and requires more fuel. These effects ripple through the design and are more pronounced for some parameters than others. Sensitivity analyses identify which parameters may affect the design disproportionately.

B. PERFORMANCE REQUIREMENTS

1. Discussion

The next step in the conceptual design process is to meet the various performance requirements, making a determination of the required thrust/weight ratio and the best wing loading. Knowing take-off weight, thrust/weight ratio, and wing loading, the student is able to make a preliminary engine selection and size the wing.

The analysis provided by this section of the design program determines the acceptable combinations of thrust/weight ratio and wing loading for five performance requirement areas. These areas are displayed to the student in the Chapter Three menu as shown in Figure 3.1.

```
CHAPTER III. PERFORMANCE REQUIREMENTS
*************************************

     MATCHING / MAIN MENU

 1. Introduction

 2. Take-off distance
 3. Climb requirements
 4. Cruise requirements
 5. Maneuvering requirements
 6. Landing requirements

 7. Review/store data
 8. Recover previous data
 9. Graph results

10. Return to Chapter Selection
```

Figure 3.1 Performance Requirements Menu

In any set of specifications, certain performance requirements will be more demanding than others and hence "drive" the design. By graphing the various combinations of thrust/weight ratios vs. wing loadings for each of the requirements, the student can select an appropriate match of these parameters (i.e., one which will meet the performance specifications in each category). The optimum combination is a trade-off favoring the highest qualifying wing loading and the lowest allowable thrust/weight ratio.

Figure 3.2 shows a sample graph of performance requirements for a light-weight fighter design. The design program has the capability to summarize the results of the five performance categories and produce such a graph. It can be seen from this graph that this design is "driven" by the cruise and maneuver specifications. An appropriate wing-loading would be 63 psf with a thrust/weight ratio of 0.83. A higher wing loading could be chosen if a more powerful "off-the shelf" engine were to be used. For example, a wing loading of 70 psf would be acceptable if thrust/weight were increased to 0.90. Note also, that the landing requirement places an upper limit on acceptable wing loading since the aircraft's approach speed cannot be reduced by increasing thrust to weight ratio. Depicting all performance results on a single graph rapidly reveals the locus of acceptable combinations that might otherwise be obscured.

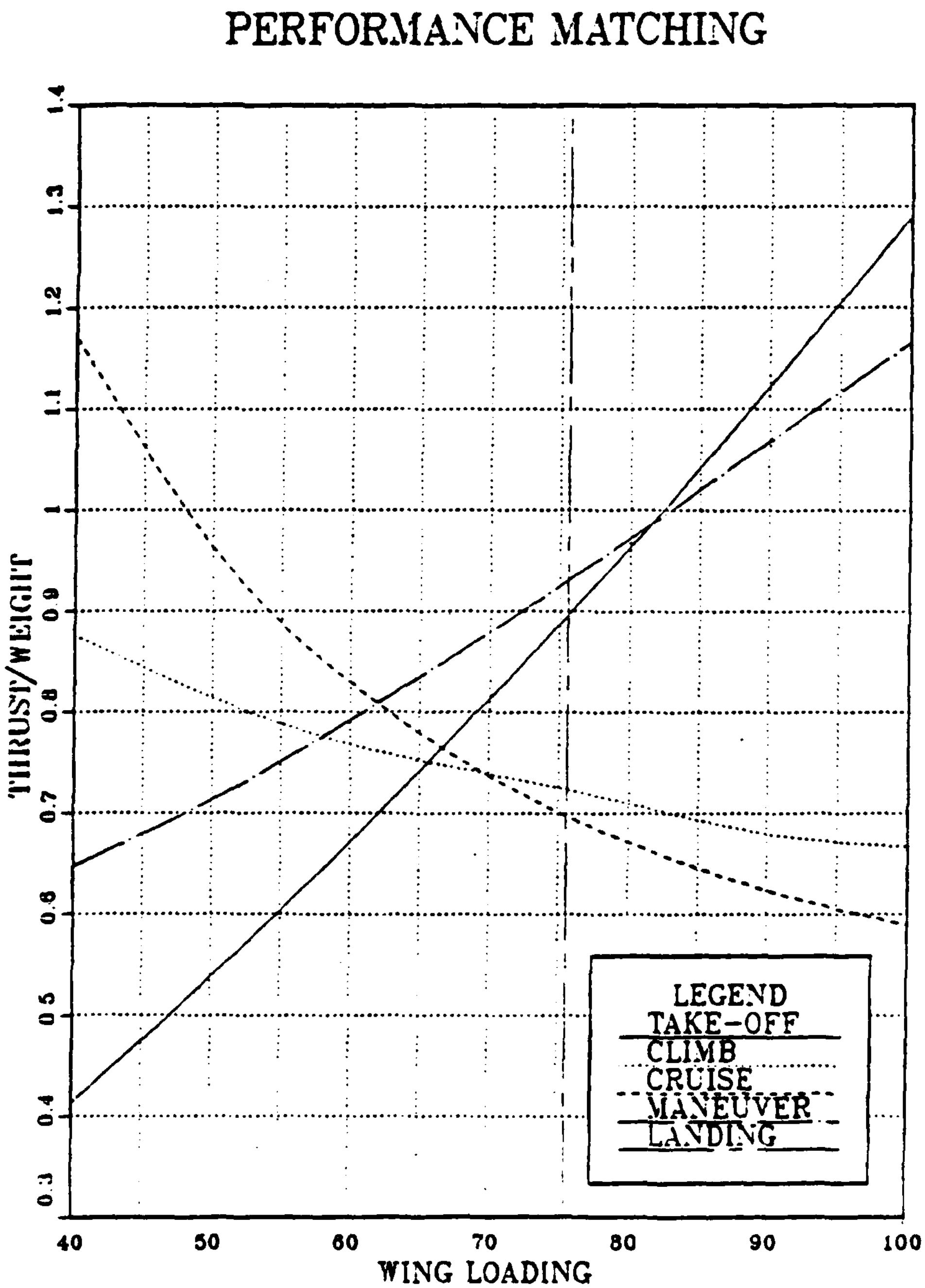

Figure 3.2 Performance Matching

2. Take-off Distance

The following relationship was used to determine the acceptable wing loading and thrust/weight combinations:

$$STO = \frac{(20.9)\,(W/S)}{(CL_{max})(sigma)(T/W)} + (87)\left[\frac{(W/S)}{(sigma)\,(CL_{max})}\right]^{.5} \qquad (3\text{-}14)$$

where STO = take-off distance
sigma = density ratio
T/W = thrust/weight
W/S = wing loading
CL_{max} = maximum lift coefficient in the landing configuration

Solving for T/W required gives:

$$T/W = \frac{\dfrac{(20.9)\,(W/S)}{(sigma)\,(CL_{max})}}{STO - (87)\left[\dfrac{(W/S)}{(sigma)\,(CL_{max})}\right]^{.5}} \qquad (3\text{-}15)$$

This equation is solved for T/W for various wing loadings, holding the remaining input parameters constant. The design program calculates, lists, and stores the acceptable combinations of thrust/weight ratios and wing loadings for a wing loading range of 30 to 125 psf.

3. Climb Performance

The performance specifications call for the aircraft to climb to a specified altitude within a specified length of time. Determination of the acceptable combinations of wing loadings and thrust/weight ratio for this specification requires knowledge of the following three factors:

a. thrust available, and its variation with altitude

b. local pressure,and its variation with altitude

c. Gamma, CDO, aspect ratio, and e.

From basic performance theory [2] it can be shown that if thrust is independent of velocity, the maximum rate of climb for a particular altitude occurs at a Mach number which satisfies the following relationship:

$$M^2 = \frac{T}{6A} + \frac{T}{6A} + \frac{B}{3A} \quad ^{.5} \qquad (3\text{-}16)$$

where $A = (\gamma/2)(p)(CDO)(S)$

$$B = \frac{(2K)(W \cos \theta)^2}{(\gamma)(p)(S)} \qquad (3\text{-}17)$$

M = mach
T = thrust
p = pressure
S = wing area
K = 1 /[(π)(AR)(e)]
W = aircraft weight
S = wing surface area
θ = climb angle
γ = C_p / C_v

Knowing the climb mach and climb angle yields the climb rate. The process becomes iterative, however, because the climb angle (θ) is initially unknown. Nevertheless, the required angle can be found using the following relationship:

$$\sin(\theta) = \frac{\text{Thrust available - drag}}{\text{Weight}} \qquad (3\text{-}18)$$

The solution begins by assuming a moderate climb angle (i.e, 10 degrees). The calculation of A,B,M, and drag follow in order. The angle is revised, and the steps are repeated. This procedure converges rapidly, and good results are obtained within four iterations.

Another complexity arises from the variations of pressure and thrust with altitude. As the aircraft climbs, the temperature decreases until reaching the tropopause. The pressure also decreases continuously with increasing altitude. The result of climbing is an interplay between pressure and temperature variations, giving a decreasing thrust. "An increase in altitude then causes the engine air flow mass to decrease in a manner very nearly identical to the altitude density ratio. Actually, the variation of thrust with altitude is not quite as severe as the density variation because favorable decreases in temperature occur. The decrease in temperature will provide a relatively greater

combustion gas energy and allow a greater jet velocity. The increase in jet velocity somewhat offsets the decrease in mass flow". [3:119]

The variation of thrust with altiude can be approximated as:

Thrust = (thrust at sea level) (delta) (1/TMPR)

where delta = pressure ratio (3-19)
TMPR = temperature ratio.

The net result of changing pressure and thrust is a continuously changing climb angle and climb rate as the aircraft climbs. At this point a computer solution becomes virtually a requirement. The design program provided by this thesis computes an optimum climb mach, climb angle, and climb rate every thousand feet until reaching the specified altitude. If the total time required is not within 0.2 seconds of the specified time, the process is repeated with an adjusted take-off thrust. This procedure continues until the minimum acceptable take-off thrust/weight ratio is found for a particular wing loading. The process is repeated for twenty wing loadings, from 30 to 125 psf. The final results are then displayed in tabular and graphical forms.

4. Cruise Performance

The third performance area evaluated was cruise performance, (i.e., required cruise speed or required level

flight speed). The specifications require that the aircraft be able to cruise at a specific altitude and airspeed. At maximum cruise speed, the following equations are simultaneously satisfied:

$$\text{Thrust} = \text{drag} = (CD)\ (q)\ (S) \qquad (3\text{-}20)$$

where CD = aircraft drag coefficient
q = dynamic pressure
S = wing surface area

and

$$\text{Weight} = (CL)\ (q)\ (S) \qquad (3\text{-}21)$$

If a parabolic drag polar is assumed, the thrust required equation may be written as:

$$TR = (CDO)\ (q)\ (S) + \frac{(CL^2)\ (q)\ (S)}{(\pi)\ (AR)\ (e)} \qquad (3\text{-}22)$$

where TR = thrust required.

Dividing by weight:

$$\frac{TR}{W} = \frac{(CDO)\ (q)}{(W/S)} + \frac{(W/S)}{(q)\ (\pi)\ (AR)\ (e)} \qquad (3\text{-}23)$$

After computing the dynamic pressure for the specified altitude and Mach number, the design program constructs a table of the relations between T/W and W/S which satisfies the maximum cruise speed requirements. These

results are then included with the other performance results on the performance matching graph.

5. Maneuvering

The specification for maneuvering perfomance is typically defined in terms of a sustained G-load at a particular mach and altitude. The sustained maneuvering capability of an aircraft depends strongly on its maximum lift coefficient and on its installed thrust. The design program computes the thrust/weight ratio required to achieve the specified turn performance at various wing loadings. The thrust/weight ratio and wing loading parameters are then analayzed to see if the required lift coefficient is reasonable. As with the other performance results, these relationships are tabulated, stored, and then plotted on the performance requirements matching graph. The procedure for making these calculations is outlined as follows:

For equilibrium conditions it is clear that

$$(N)\ (W) = (CL)\ (q)\ (S) \tag{3-24}$$

where N is the G-load.

$$\text{Thrust} = \text{Drag} = (CDO + (K)\ (CL^2))\ (q)\ (S) \tag{3-25}$$

After dividing eqn. 3-21 by 3-22 and rearranging, it can be shown that:

$$T/W = \frac{(q)\ (CDO)}{(W/S)} + \frac{(K)\ (W/S)\ (N)^2}{(q)} \qquad (3\text{-}26)$$

where T/W = thrust/weight required
N = G-load specified
K = $1/[(\pi)\ (AR)\ (e)]$
q = dynamic pressure.
G = gravitational constant

It can also be shown that the specification of a G-load and a velocity at a particular altitude defines a turn rate according to the following relationship:

$$\text{turn rate} = \frac{G}{V} \{ (N^2 - 1)^{.5} \} \qquad (3\text{-}27)$$

where Turn rate is measured in radians/sec,
V = velocity
N = specified G-load.
G = gravitational constant

The computed turn rate is displayed in the data summary since it is a primary performance comparison figure for tactical aircraft.

As a second option for maneuvering analysis, the program allows the designer to check whether the lift coefficient required to meet the previous maneuvering specifications is within reasonable limits. The previous computations for wing loading and thrust/weight ratio placed no limitations on CL. As a cross check, this section displays

the maneuvering CL associated with each wing loading to allow the student to ensure that realistic limits are observed.

The inputs required to compute CL are:

(1) turn rate
(2) G-load
(3) altitude
(4) wing loading.

The computations for CL proceeds as follows:

$$\text{Velocity (fps)} = G * \frac{(N^2 - 1)^{.5}}{(\text{turn rate})} \qquad (3\text{-}28)$$

$$q = .5\ (\text{density})\ (\text{velocity})^2 \qquad (3\text{-}29)$$

$$(N)\ (W) = (q)\ (S)\ (CL) \qquad (3\text{-}30)$$

$$CL = (W/S)\ (N/q) \qquad (3\text{-}31)$$

where CL = coefficient of lift
W/S = wing loading
N = specified G-load.

6. Landing Distance

The final performance calculations were made for the landing distance requirements. Before beginning the calculations, however, it is particularly important to clearly specify the particular definition of landing distance being used, since there are several common definitions. For the purposes of this section the definition that was

programmed for analysis was developed by Jan Roskam[4]. This procedure assumes a particular ratio of ground roll to total landing distance. Additionally, the ratio of total landing distance to field length is specified by FAR Regulations to be the following relations:

$$SL = 1.9 * SLG \tag{3-32}$$

$$SFL = SL / 0.6 \tag{3-33}$$

where SLG = landing ground run
SL = total distance during landing
SFL = field length.

From landing performance analyses, a relationship can be made between the required field length and the approach speed:

$$VA1 = 1.8367 \{ SFL)^{.5} \} \tag{3-34}$$

where VA1 is the reference approach speed in knots.

This relationship assumes that for safety considerations, the approach speed is 1.3 times the stall speed; however, since an approach safety factor of less than 1.3 is generally used by tactical aircraft, the computations must be adjusted when considering their non-standard approach speeds. (Note: The effect of the reduced stall margin used by tactical aircraft is to decrease the landing distance by the square of the approach speed ratio. This adjustment is made in eqn. 3-36).

For performance chart graphing it is necessary to determine the maximum wing loading which would allow the aircraft to meet the landing distance specifications. The inputs required by the program to do this are:

(1) total landing distance, SL
(2) density ratio
(3) CLmax
(4) approach safety factor, ASF.

The calculations proceed as follows:

$$SFL = SL/0.6 \tag{3-35}$$

$$VA2 = \{ (VA1^2)\ (1.3/ASF)^2 \}^{.5} \tag{3-36}$$

$$VS1 = VA2/ASF \tag{3-37}$$

$$VS2 = (VS1)\ (6076/3600) \tag{3-38}$$

$$(W/S)_L = (1/2)\ (density)\ (VS2)^2\ (CLmax) \tag{3-39}$$

where SFL = landing field length
SL = total landing distance
VA1 = reference approach speed, knots
VA2 = adjusted approach speed, knots
VS1 = adjusted stall speed, knots
VS2 = adjusted stall speed, feet/sec
ASF = approach safety factor
$(W/S)_L$ = wing loading, landing.

The landing wing loading, $(W/S)_L$, is then normalized to the take-off wing loading for plotting on the performance requirements graph by dividing $(W/S)_L$ by the weight fraction determined during the mission analysis. (This weight fraction

is automatically recalled from the data files for the convenience of the student). Computations are made as follows:

$$(W/S)_{TO} = (W/S)_L \; / \; (WL/WTO) \tag{3-40}$$

where (WL/WTO) = landing weight/take-off weight
$(W/S)_{TO}$ = wing loading, take-off
$(W/S)_L$ = wing loading, landing.

It should be noted that the landing requirement serves to fix an upper limit on the acceptable wing loading. This limit cannot be increased by the addition of thrust, as with the other performance parameters, since thrust is not a limiting factor in reducing the approach speed.

Finally, it should be emphasized that the thrust/weight ratio and wing loading relationships for all performance categories must be normalized to a common reference condition if they are to be plotted on the same graph. This reference condition is typically take-off wing loading and take-off thrust/weight. For example, if the aircraft were expected to land at 80% of its take-off weight, the wing loading computed for the landing requirement would be 80% of the reference take-off wing loading. The design program allows entry of these normalizing ratios for both wing loading and thrust/weight parameters.

IV. ASPECT RATIO OPTIMIZATION

A. DISCUSSION

Selection of the optimum aspect ratio is a major factor in aircraft design. Equations 4-1 and 4-2 show that the drag coefficient and the drag itself are reduced by using a large aspect ratio.

$$CD = CDO + \frac{CL^2}{(\pi)(AR)(e)} \qquad (4-1)$$

$$Drag = (CD)(q)(S) \qquad (4-2)$$

Since aspect ratio is defined as b^2/S it can be seen that for a given wing area (S), a large aspect ratio means a large span.

> From a pure drag standpoint, the larger the span can be, the better the airplane design will be. However, a large span means larger bending moments in the wing structure because the lift loads are acting farther from the root chord of the wing. Furthermore, a large span with a fixed area means shorter wing chords all along the span and, therefore, thinner wings. The wing acts as a beam, and a shallow beam requires heavier material on the top and bottom of the structure to withstand a given bending moment. Thus a high-aspect-ratio wing has a heavier structure. The higher wing weight raises the average flying weight and therefore, increases the drag, counteracting some of the aerodynamic drag gain. Also a thinner wing with a longer span has less internal volume for fuel. The most efficient wing depends on the range, design cruise speed, and the cost of fuel. [5:183]

For purposes of the design program, the selection criteron used for aspect ratio optimization was minimum take-off weight. In other words, a particular aspect ratio was considered to be better then another if it resulted in a lower take-off weight.

The analysis calculates a wing weight penalty incurred for increased aspect ratio. This structural weight penalty is countered by fuel weight savings. The fuel savings result from an improved L/D, since the drag coefficient decreases as aspect ratio increases. Therefore, one can anticipate a decrease in fuel weight requirements as aspect ratio increases.

The design program analyzes the above problem and performs two variations of this idea. The menu from Chapter III of the design program displays these methods as shown in Figure 4.1.

Chapter III. ASPECT RATIO OPTIMIZATION

1. Introduction

2. Fixed Mach Method
3. Variable Mach Method

4. Return to CHAPTER SELECTION

Figure 4.1 Aspect Ratio Optimization Menu

B. FIXED MACH METHOD

The "Fixed-Mach" method of aspect ratio optimization computes the required take-off weights for aircraft of varying aspect ratio while flying the mission at a specified Mach number. The following conditions are imposed:

1. Aircraft flies mission profile as specified in Chapter II of the design program
2. Aircraft incurs a fixed weight adjustment based on the deviation of the wing weight at the chosen aspect ratio from a specified reference aspect ratio
3. Wing loadings for each phase are derived from the specified take-off wing loading using the weight fraction calculated previously. For example the average wing loading during the cruise-out phase would be:

$$(W/S)_{cruise} = (W/S)_{TO} \frac{(W2)}{(WTO)} \frac{(W3)}{(W2)} \frac{(1 + W4/W3)}{(2)} \qquad (4\text{-}3)$$

where $(W/S)_{cruise}$ = mid cruise wing loading
$(W/S)_{TO}$ = take-off wing loading

4. L/D inputs for cruise and loiter portion are computed for each aspect ratio using the assumption of a common CDO, wing-loading, and efficiency factor as shown in equations 4-6 through 4-8

$$q = (1/2)\ (P)\ (M^2) \qquad (4\text{-}4)$$

where M = specified cruise mach
P = pressure

$$CL = (W/S)_{cruise} / (q) \qquad (4\text{-}5)$$

$$CD = CDO + (K)(CL^2) \tag{4-6}$$

$$(L/D)_{cruise} = (CL) / (CD) \tag{4-7}$$

$$(L/D)_{loiter} = (L/D)_{max} = 1/(2)[(CDO)(K)]^{.5}. \tag{4-8}$$

After the fixed weight adjustment and L/D inputs are evaluated, the program "flies" the mission profile and computes the take-off weight for twenty-six aspect ratios ranging from 2.5 to 5.0. Again optimum aspect ratio for purposes of this analysis is considered to be the the one producing the minimum take-off weight. This optimization balances structural weight penalties against fuel savings.

C. VARIABLE MACH METHOD

The second method assumes that each aspect ratio airplane is flown at its own optimum speed. An upper limit of 0.9 mach is imposed to minimize compressibility considerations, which have been ignored. For purposes of this section, the optimum speed is defined as the one which minimizes the fuel burn for the phase.

The optimum speed for the cruise leg may be shown to be the one which maximizes the multiplication factor in the Brequet range equation in expression 4-9:

$$[\{(V) / (SFC)\} (L/D)]_{max} \tag{4-9}$$

where SFC = the specific fuel consumption
V = the aircraft cruise velocity.

For each aspect ratio, the design program determines the optimum cruise velocity (V) by maximizing relation 4-9. In computing this maximum, it is assumed that specific fuel consumption varies linearly with velocity. This variation is defined by two reference points provided by the user. The cruise L/D used in equation 4-9 varies with velocity according to the following equations:

$$q = 1/2\ (\rho)(V^2) \tag{4-10}$$

$$CL = (W/S)\ /\ (q) \tag{4-11}$$

$$CD = CD0 + (K)(CL^2) \tag{4-12}$$

$$(L/D)_{cruise} = CL\ /\ CD. \tag{4-13}$$

Note: The analysis was originally performed with the assumption that specific fuel consumption was independent of mach. This assumption led to outputs of excessively low aspect ratios by historical standards. Further investigation revealed that for the typical modern fighter engine of low to medium bypass ratio the specific fuel consumption (SFC) changes significantly with mach. For example, the particular engine studied in detail showed an SFC of 0.78 at mach 0.5 and an SFC of 0.88 at a mach of 0.9. The dependence of SFC on mach is a strong function of engine bypass ratio. As engine bypass ratio increases, the SFC varies even more significantly with mach.

The program then calculates the loiter L/D and loiter SFC. The assumption is made that the aircraft will loiter at $(L/D)_{max}$. SFC computations parallel those described for cruise.

The results of optimizing cruise and loiter performance show that as aspect ratio increases, optimum mach decreases and fuel efficiency increases. The relative magnitude of these variations determines the aspect ratio associated with minimum WTO.

Finally, the program results are listed in tabular output to allow plotting aspect ratio against WTO. The designer should note whether the curve for minimum WTO is flat or sharp. The shape of this curve affects the amount of flexibility the designer may have in selecting an aspect ratio.

It should be noted that the criteria of minimizing WTO is only one of many possible methods which might be considered in calculating the "optimum" aspect ratio. For a naval fighter/attack aircraft, the need to minimize deck space requirements may favor chosing a lower aspect ratio than that which produces minimum WTO. Nevertheless, a decision to choose a low aspect ratio for a twin engine aircraft must be tempered by the requirements for acceptable single engine performance.

For twin engine aircraft which must be able to climb with only one engine operative after one engine fails, a higher aspect ratio may be chosen to improve low speed climb performance even though it is greater than the optimum for cruising flight. In low speed climbing flight the induced drag may be 75% of the total drag, and aspect ratio has an enormous effect on performance. [5:184]

A sample output for method #1 (fixed mach) is shown in Figure 4.1. Note: minimum WTO occurs at an aspect ratio of 3.1 for this example.

SUMMARY OF ASPECT RATIO OPTIMIZATION
****

```
W/S (T.O.) =  80   CD01 = 0.025  M1 = 0.80  ALT1 = 30000  Q1 = 282
AR (ref)   = 3.4   CD02 = 0.020  M2 = 0.78  ALT2 = 36000  Q2 = 202
```

AR	l/d1	l/d2	l/d3	WWP	WTO	AR	l/d1	l/d2	l/d3	WWP	WTO
****	****	****	****	***	*****	****	****	****	****	***	*****
2.5	7.25	8.74	8.86	-870	59210	3.8	8.09	10.26	10.93	370	59140
2.6	7.34	8.89	9.04	-770	59023	3.9	8.14	10.35	11.07	461	59264
2.7	7.42	9.03	9.21	-671	58880	4.0	8.19	10.44	11.21	552	59397
2.8	7.49	9.16	9.38	-573	58777	4.1	8.23	10.52	11.35	642	59539
2.9	7.57	9.29	9.54	-476	58710	4.2	8.27	10.60	11.49	732	59690
3.0	7.64	9.41	9.71	-379	58674	4.3	8.31	10.68	11.62	821	59848
3.1	7.70	9.53	9.87	-284	58665	4.4	8.35	10.76	11.76	910	60014
3.2	7.77	9.65	10.03	-188	58679	4.5	8.39	10.84	11.89	998	60185
3.3	7.83	9.76	10.18	-94	58715	4.6	8.42	10.91	12.02	1086	60361
3.4	7.89	9.86	10.34	-0	58770	4.7	8.46	10.98	12.15	1173	60544
3.5	7.94	9.97	10.49	93	58841	4.8	8.49	11.05	12.28	1261	60731
3.6	7.99	10.07	10.63	186	58928	4.9	8.53	11.12	12.41	1347	60922
3.7	8.05	10.16	10.78	278	59028	5.0	8.56	11.19	12.53	1434	61117

Press enter to continue.

Figure 4.1 Aspect Ratio Optimization

V. WING GEOMETRY

A. DISCUSSION

Chapter Five of the design program solves wing geometry equations. Options are presented to the user as shown in Figure 5.1.

```
CHAPTER V. WING GEOMETRY
************************

1. Introduction

2. Sweep Angle: leading edge
3. Sweep Angle: 1/4 chord
4. Wing Area
5. Span
6. Root and Tip Chord
7. Mean Aerodynamic Chord  and
   Center of Pressure

8. Return to CHAPTER SELECTION
```

Figure 5.1 Wing Geometry Selection Menu

All calculations above use the conventional aeronautical definitions and relationships. This chapter provides a convenient format for geometric calculations which are frequently repeated. See Figures 5.2 and 5.3 for a listing of wing geometry formulas.

WING GEOMETERY FORMULAS: Part 1

****

Section 2: Sweep Angle Leading Edge, degrees ($Sweep_{LE}$)

Given: design mach (DM)

Assymption: Supersonic wing with subsonic leading edge. Wing swept five degrees behind the mach line.

Formula: $$Sweep_{LE} = 95 - \tan^{-1}\left[\frac{1}{(DM^2 - 1)}\right] \quad (5\text{-}1)$$

Section 3: Sweep Angle 1/4 chord ($Sweep_{c/4}$)

Given: a. Sweep angle leading edge ($Sweep_{LE}$)
b. Taper ratio (L)
c. Aspect Ratio (AR)

Assumption: trapezoidal wing

Formula: $$\tan(Sweep_{c/4}) = \tan(Sweep_{LE}) - \left(\frac{(1 + L)}{(AR)\ (1 - L)}\right) \quad (5\text{-}2)$$

Section 4: Wing Area

Given: a. Take-off weight (WTO)
b. Take-off wing loading (WSTO)

Assumption: none

Formula: $$S = WTO / WSTO \quad (5\text{-}3)$$

Figure 5.2 Wing Geometry Formulas: Part 1

WING GEOMETRY FORMULAS: Part 2

Section 5: Span (b)

Given: a. Aspect ratio (AR)
b. Wing surface area (S)

Assumption: trapezoidal wing

Formula: $b = \{ (AR)\ (S) \}^{.5}$ (5-4)

Section 6. Root Chord (CR) and Tip Chord (CT)

Given: a. Wing surface area (S)
b. Wing span (b)
c. Taper ratio (L)

Assumption: trapezoidal wing

Formulas:

$$Cr = \frac{(2)\ (S)}{(b)\ (1 + L)} \qquad (5\text{-}5)$$

$$Ct = (Cr)\ (L) \qquad (5\text{-}6)$$

Section 7: Mean Aerodynamic Chord (MAC) Spanwise distance to Center of Pressure (Ybar)

Given: a. Wing span (b)
b. taper ratio (L)
c. Root chord (Cr)

Assumption: trapezoidal wing

Formulas:

$$MAC = \left[\frac{(2)\ (Cr)}{(3)}\right] \left[\frac{(1 + L + L^2)}{(1 + L)}\right] \qquad (5\text{-}7)$$

$$Ybar = \left[\frac{(b)}{(6)}\right] \left[\frac{[1 + (2)(L)]}{(1 + L)}\right] \qquad (5\text{-}8)$$

Figure 5.3 Wing Geometry Formulas: Part 2

A sample of inputs and results for item 7, (Mean Aerodynamic Chord and Center of Pressure), is presented in Figure 5.4.

MEAN AERODYNAMIC CHORD AND CENTER OF PRESSURE
****

```
        Note: (previous values)     Wing span   = 35.60
                                    Taper ratio = 00.24
                                    Root chord  = 12.80

 1. Input wing span?           40.0
 2. Input taper ratio?          0.2
 3. Input root chord?          15.0

   COMPUTATION RESULTS
   -------------------

MEAN AERODYNAMIC CHORD          = 10.33 ft
DISTANCE TO CENTER OF PRESSURE =  7.78 ft
```

Figure 5.4 Sample Wing Geometry Calculation

VI. FUSELAGE LENGTH

Chapter Six of the design program uses regression formulas to predict fuselage length. These regression formulas are based on empirical data relating fuselage length to take-off weight. This simple relation was chosen for the design program because of the excellent correlation obtained with data for modern tactical aircraft. An alternate method which sizes the fuselage using the volume requirements of internal components, was rejected because the greatly increased "bookkeeping" showed no payoff in increased accuracy.

The first regression formula uses the following terms:

$$\text{fuselage length} = (A)\,(WTO)^{B} \qquad (6\text{-}1)$$

where A and B are defined as follows:

		A	B
(1)	jet fighter	0.83	0.39
(2)	jet trainer	0.79	0.41

The second formula is used for supersonic aircraft only:

$$\text{fuselage length} = 41 + (0.0043)\,(WTO) \qquad (6\text{-}2)$$

Figure 6.1 presents a listing of results for eight modern fighter aircraft. (The data source for the take-off weights is Aviation Week [6].)

AIRCRAFT	GROSS WEIGHT (TAKE OFF)	ACTUAL LENGTH	PREDICTED LENGTH METHOD 1	PREDICTED LENGTH METHOD 2
1. F-4S	56,000	58.3	59.0	60.0
2. F-5E	24,722	47.4	42.9	49.4
3. F-14A	59,714	62.7	60.5	61.3
4. F-15C/D	69,000	63.8	64.0	64.5
5. F-16C	24,537	47.6	42.8	49.3
6. F/A-18	51,900	56.0	57.3	58.6
7. F-111	100,000	75.5	74.0	75.0
8. F-21A	32,413	51.3	47.7	52.0

Figure 6.1 Fuselage Length Results

VII. VERTICAL TAIL DESIGN

Chapter Seven of the design program solves the iterative problem of obtaining a particular tail volume coefficient. The required input parameters are listed as follows:

1. desired tail volume coefficient
2. fuselage length
3. CG position on fuselage
4. wing sweep
5. wing aspect ratio
6. wing taper ratio
7. wing surface area
8. CG position as a fraction of MAC
9. distance of tail from end of fuselage
10. tail sweep
11. tail taper ratio.

The program calculates the size requirements for a vertical tail meeting the specified tail volume coefficient subject to the above input conditions. Calculations begin by determining the location of the center of pressure for the wing. The program then selects an initial surface area for the vertical tail shape defined by the user. (Note: the user's inputs of the vertical tail sweep, aspect ratio, and taper ratio, have fixed the basic planform shape of the vertical tail). The trailing edge of this vertical tail is positioned at the location previously defined by the user (item 9). All parameters necessary to calculate a vertical tail volume coefficient (C_{V_T}) are then available. The

calculations are performed, and a comparision is made with the desired specification value for C_{VT}. Through an iterative process, the tail surface area is adjusted, (while maintaining all input parameters), until the specified value for C_{VT} is achieved. The solution values for the tail and wing geometries are then summarized for the user and presented as shown in Figure 7.1.

The tail volume coefficient is defined as shown by equation 7-1:

$$C_{VT} = \frac{(Lvt)\ (Svt)}{(bw)\ (Sw)} \qquad (7\text{-}1)$$

where Lvt = length between the center of pressure of the wing and the center of the vertical tail
Svt = surface area of vertcal tail
bw = wing span
C_{VT} = coefficient of vertical tail.

TABLE OF CHAPTER SEVEN RESULTS
****

	WING	TAIL
1. Surface area	600.00	99.71
2. Sweep (degrees)	45.00	45.00
3. Aspect ratio	3.20	1.50
4. Span (ft)	43.82	8.65
5. Taper ratio	0.20	0.50
6. Leading edge position	20.19	41.62
7. Trailing edge position	43.01	57.00
8. Root chord length	22.82	15.38
9. center of pressure	32.64	48.46
10. sweep of 1/4 chord	39.09	30.96
11. mean aerodynamic chord	15.72	11.96
12. Y bar	8.52	3.85

TAIL VOLUME COEFFICIENT = 0.060 FUSELAGE LENGTH = 60.00

A/C CENTER OF GRAVITY = 35.00 TAIL LENGTH Lvt = 15.81

A/C CG POSITION, %MAC = 40.00 BOATTAIL LENGTH = 3.00

Figure 7.1 Tail Sizing Results

VIII. DETERMINING STRUCTURAL WEIGHTS

Chapter Eight of the design program solves empirical weight estimation formulas for six major aircraft components, which are used to refine WTO now that more is known about the design. The chapter menu is presented to the user as shown in Figure 8.1.

```
CHAPTER VIII. DETERMINING STRUCTURAL WEIGHTS (WS)
*************************************************

1. Introduction

2. Wing
3. Horizontal Tail
4. Vertical Tail
5. Fuselage
6. Main Landing Gear
7. Nose Landing Gear

8. Return to CHAPTER SELECTION
```

Figure 8.1 Chapter Eight Menu

Following selection of a particular option, the user is presented with a component weight menu similar to the example in Figure 8.2. The program then calculates an estimated component weight based upon inputs to requested parameters. (See Sample of Input Pararemters, Figure 8.3.)

CHAPTER 8.2: WING WEIGHT ESTIMATE
****

1. List input parameters and current values.
2. Input a new set of values for parameters.
3. Change a single parameter value.
4. Store / Recover parameter data

5. Return to STRUCTURAL WEIGHTS MENU

Figure 8.2 Sample Component Weight Menu

******** INPUT A NEW SET OF PARAMETERS *********

1. Input K.DW (.768 delta wing, 1.0 non-delta wing)?

2. Input K.VS (1.19 variable sweep, 1.0 fixed wing)?

3. Input K.FOLD (1.1 with fold, 1.0 no fold)?

4. Input W.DG (Design gross weight - lbs)?
   (approximately (WE + WF))

5. Input N.Z (Ultimate load factor)?

6. Input S (Gross wing area - ft sq)?

7. Input AR (Wing aspect ratio)?

8. Input T.CR (wing thickness divided by root chord)?

9. Input Lambda (Wing taper ratio)?

10. Input GAMMA (Wing sweep angle at 25% chord)?

11. Input S.CS (Area - wing mounted control surfaces)?
    (approximately 25% of wing area)

Figure 8.3 Sample of Input Parameters

The user is given various options for manipulating the component inputs. A particularly useful feature of the program is the ability to vary a single parameter through a specified range to observe the effects upon the component weight. For example, variation of aspect ratio for a particular wing produces the results shown in Figure 8.4.

Note: Reference value of parameter 7 = 3.4

N	PARAMETER 7	WEIGHT	CHANGE IN WEIGHT
***	***********	******	***************
1.	2.00	2430	-1256
2.	2.20	2619	-1067
3.	2.40	2804	-882
4.	2.60	2986	-700
5.	2.80	3165	-521
6.	3.00	3341	-345
7.	3.20	3515	-171
8.	3.40	3686	0
9.	3.60	3855	169
10.	3.80	4023	336
11.	4.00	4188	502
12.	4.20	4351	665
13.	4.40	4513	827
14.	4.60	4673	987
15.	4.80	4832	1146
16.	5.00	4990	1303

Figure 8.4 Sample of Parameter Variation

Note: The change in estimated wing weight induced by changes in aspect ratio (as demonstrated above) is the source of the weight adjustements used for aspect ratio optimization in Chapter Seven.

In obtaining an expression for the particular component the following technique was used by Vought:

> The general approach was first to develop an analytical expression for the component under investigation. An exponential equation was written which contained the same terms as the analytical expression. (Theoretical expression limits were established by investigation of the analytical expression). A least squares curve fitting process using statistical data was used to determine the values of the exponents in the exponential equation. Calcuated weight derived from the exponential equation was plotted vs. the actual component weights. Equations were selected both on the form and plotted results. [7:1-2]

The regression formulas used in calculating the component weights are listed in Appendix A.

IX. REFINED ESTIMATE OF WTO

A. DISCUSSION

Chapter Nine of the design program uses the combined weight of six major components (WS) to make a refined estimate of take-off weight (WTO). The following components are used for this estimate:

1. wing
2. horizontal tail
3. vertical tail
4. fuselage
5. main landing gear
6. nose landing gear.

A strong correlation was found to exist between the weight of these six components (WS) and an aircraft's empty weight (WE). This chapter uses this correlation and mission data from Chapter TWO to estimate WTO.

B. METHODOLOGY

1. Calculation of WE from WS

The Vought Weight Estimation Manual provides a detailed listings of component weights for sixteen aircraft [7:1.3]. The weight of the group of components listed above was selected as a basis for estimating an aircraft's empty weight. For each aircraft analyzed, the total weight of the six components (WS) was plotted against its empty weight (WE). After plotting the values for all sixteen aircraft, a

least squares linear regression line was calculated to relate WE and WS (Figure 9.1). The regression analysis showed a good correlation, (97.3%), between the weight of the six components and the empty weight of the aircraft. The following linear equation was obtained:

$$WE = (1.7251)\ (WS) + 4246 \qquad (9\text{-}1)$$

where WE = aircraft empty weight
WS = aircraft "structural weight".

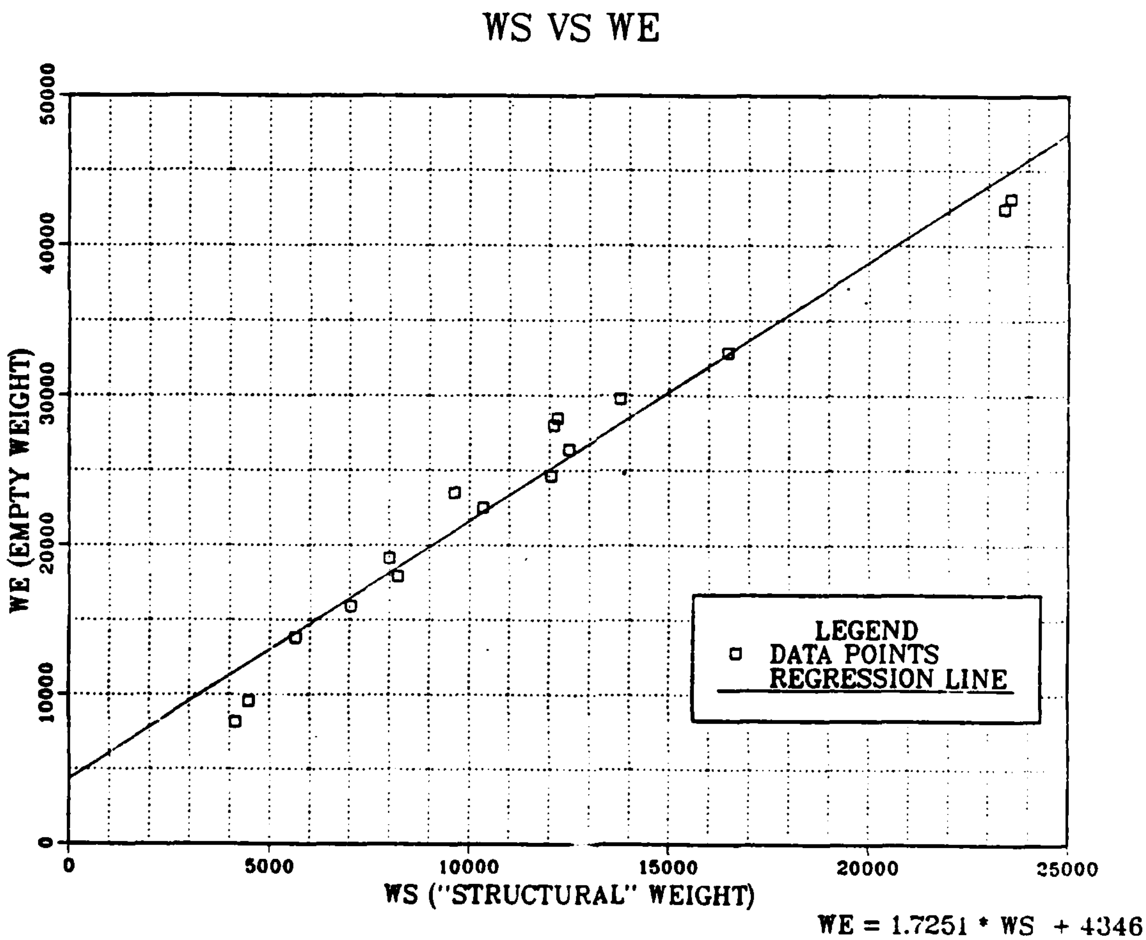

Figure 9.1 Plot of WS vs. WE

2. <u>Calculation of WDG from (WE + WF)</u>

Because the Vought component weights are developed in terms of flight design gross weight (WDG), the next step was to define a relationship involving WDG. It was further found that WDG could be related to the sum of the empty weight and fuel weight. The values were plotted as shown in Figure 9.2 to compute the following relationship:

$$WDG = (0.8933)(WE + WF) + 1026 \qquad (9\text{-}2)$$

where WDG = flight design gross weight
WE = aircraft empty weight
WF = fuel weight.

(WE+WF) VS WDG

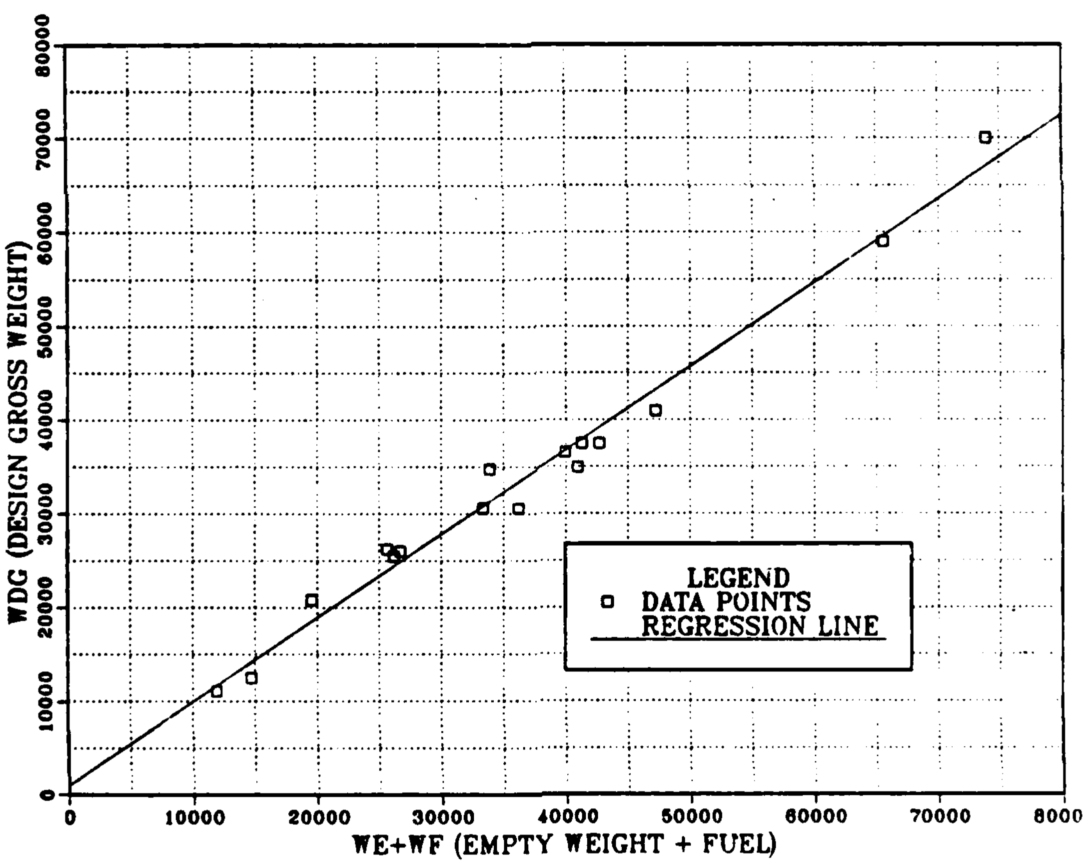

Figure 9.2 (WE + WF) vs. WDG

3. Solving for WS (Relation #1)

In Chapter Two it was shown that by using mission dependent weight fractions and a specified payload, a linear relation was obtained between WF and WTO.

$$WF = (C1)\ (WTO) + C2 \qquad (9\text{-}3)$$

From the overall weight equation, WF and WTO are related to WE.

$$WTO = WF + WE + WP \qquad (9\text{-}4)$$

Since WP is a constant, an linear expression can be written for WE also.

$$WE = (C3)\ (WTO) + C4 \qquad (9\text{-}5)$$

Combiningequations 9-3 and 9-5 provides an equation for (WE+WF) :

$$(WE + WF) = [(C1)(WTO)+C2] + [(C3)(WTO)+C4] \qquad (9\text{-}6)$$

or $$(WE + WF) = (C5)\ (WTO) + C6 \qquad (9\text{-}7)$$

Substituting eqn. 9-7 into eqn. 9-2:

$$WDG = [.8933]\ [\ (C5)(WTO)+C6\ ] + 1026 \qquad (9\text{-}8)$$

or $$WDG = (C7)\ (WTO) + C8 \qquad (9\text{-}9)$$

In order to relate WDG to WS, WTO must first be related to WS. To do this an intermediate empirical relationship between WE and WS will be used, which is the empirical results exhibited in equation (9-1).

$$WE = (1.7251)\ (WS) + 4346 \tag{9-10}$$

Recalling that:

$$WE = (C3)\ (WTO) + C4 \tag{9-11}$$

and combining eqns. 9-10 and 9-11:

$$(C3)(WTO)+C4 = (1.7251)(WS)+4346 \tag{9-12}$$

or
$$WTO = (C9)\ (WS) + C10. \tag{9-13}$$

With the relation of WTO to WS from equation 9-13, the substitution is made for WTO in equation 9-9 yielding:

$$WDG = (C11)\ (WS) + C12. \tag{9-14}$$

or
$$WS = (C13)\ (WDG) + C14 \tag{9-15}$$

Equation 9-15 is the first of two relationships for WS and WDG being sought. An example of this equation has been plotted as relation #1 in Figure 9.3.

4. Solving for WS (Relation #2)

The equation predicting component weights in Chapter Eight can each be reduced to a power form.

$$\text{component weight} = (D)\ (WDG)^{E} \tag{9-16}$$

The summation of the six conponents (WS) can be expressed as:

$$WS = \sum_{n=1}^{6} (Dn)\ (WDG)^{En} \qquad (9\text{-}17)$$

This equation is plotted as line #2 on Figure 9.3.

By combining equations 9-16 and 9-18, a single equation for WDG is obtained as follows:

$$WS = (C13)\ (WDG) + C14 \qquad \{\text{from } 9\text{-}15\} \qquad (9\text{-}18)$$

$$WS = \sum_{n=1}^{n=6} (Dn)\ (WDG)^{En} \qquad \{\text{from } 9\text{-}17\} \qquad (9\text{-}19)$$

$$[(C13)\ (WDG) + C14] = \sum_{n=1}^{n=6} (Dn)\ (WDG)^{En} \qquad (9\text{-}20)$$

$$\text{or} \qquad WDG = [\ (C15)\ (Dn)\ (WDG)^{En}\ ] + C16 \qquad (9\text{-}21)$$

When equation 9-18 and 9-19 are plotted on a common graph, the intersection of the two plots represents the common solution for WDG (Figure 9.3). (Note: The design program solves equation 9-21 through an iterative procedure.)

Finally, knowing WDG, equation 9-2 may be reversed to solve for (WE+WF). Knowing (WE+WF) and WP, the desired solution for WTO is found by recalling that:

$$WTO = (WE + WF) + WP. \qquad (9\text{-}22)$$

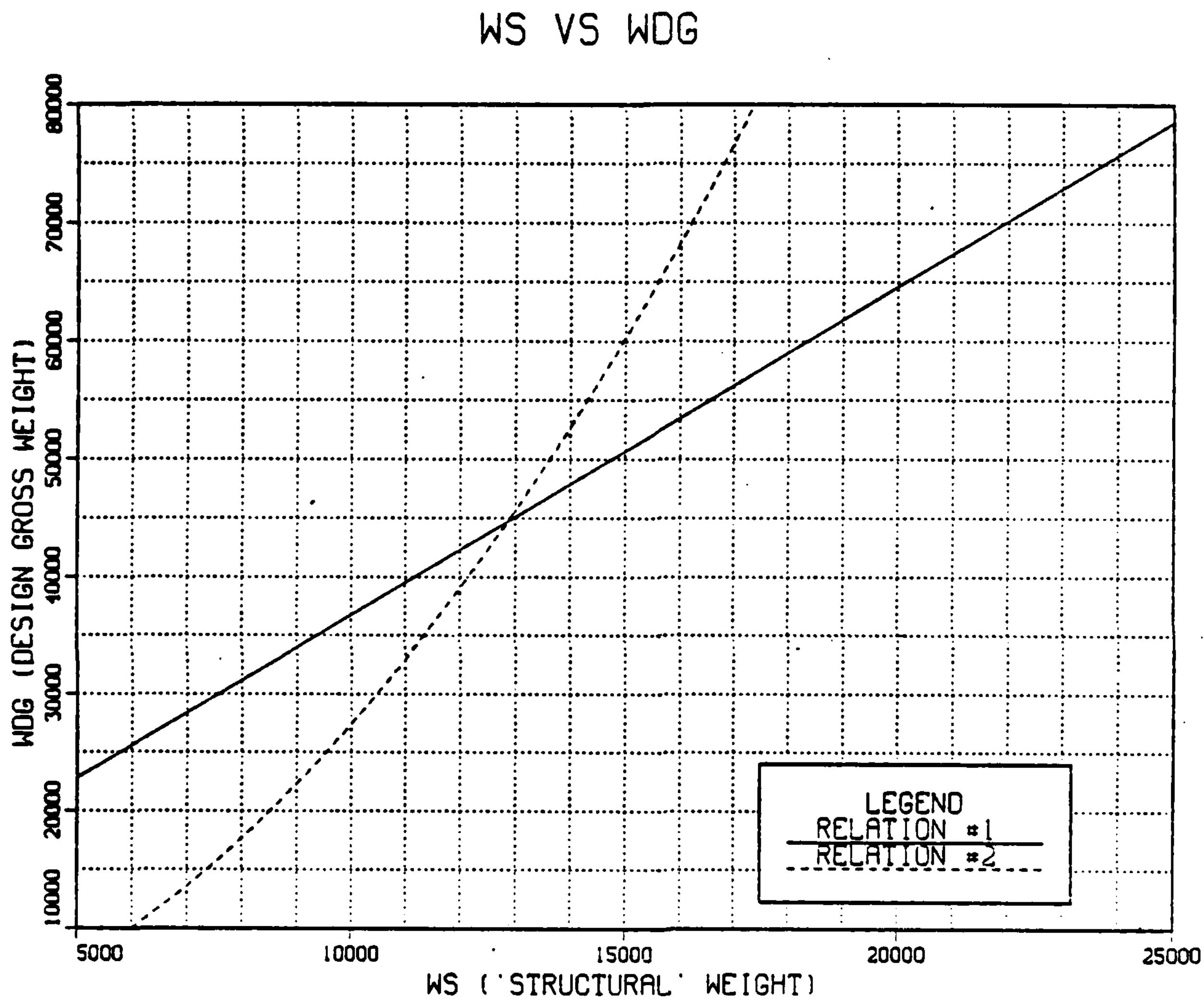

Figure 9.3 WS vs. WDG

X. CONCLUSION

The set of programs developed in this thesis have promise of materially assisting the learner grasp and come to a good understanding of the principles of conceptual aircraft design. Furthermore, it is hoped that they will improve the efficiency of learning this material by providing a tool which will conserve time for the student in phases of work which are routine and create time to cover topics heretofore not covered. This will allow the students to be exposed to aircraft design in greater depth and with greater realism.

The most precious commodity involved in the educational process at the Naval Postgraduate School is the student's time, and this set of programs is expected to make better use of that commodity by expanding significantly the meaningful information about design by officers who may well be involved in the future with the development, procurement or management of new aircraft.

The results of this thesis represent about half of the package envisioned for instruction in design; therefore, future work will continue in the same vein to cover the remaining topics needed to complete the course.

APPENDIX A
AIRCRAFT DESIGN PROGRAM USER'S GUIDE

CHAPTER ONE - INTRODUCTION

A. DISCUSSION

The computer program written for this thesis is divided into ten chapters. These chapters are addressed through a common menu called the Chapter Selection Program. (See Figure A.1).

```
**** CHAPTER SELECTION PROGRAM ****
***********************************

             CHAPTERS
             ********

 1. Introduction

 2. Preliminary Estimate of Take-off Weight
 3. Meeting Performance Requirements
 4. Aspect Ratio Optimization
 5. Wing Geometry Design
 6. Estimating Fuselage Length
 7. Tail Design
 8. Determining Structural Weights (WS)
 9. Refined Estimate of WTO Using WS

10. End Session
```

Figure A.1 Chapter Selection Program

The program is completely interactive and proceeds in stages which parallel the developments in the design course. Results are summarized at the end of individual sections. Input and output data is stored in data files for efficient

operation. These data files are written onto the diskette to provide a common data base between chapters and to provide a permanent storage for completed work. A single diskette is used for both the program and the data files for convenience of operation.

Topics of the program are discussed in detail during the Aircraft Design course. The program is intended to supplement the course as a tool to expedite completion of a significant portion of the many calculations required. Since design processes are iterative, and thus very time consuming, it is hoped that by using this program the student will be able to progress more quickly through these topics, freeing time to be exposed to additional material.

B. GETTING STARTED

After loading your system DOS, place the design diskette in drive "A". Type the command "Design" to begin program operation. If a particular program "chokes" at any time you may end operation by using "Ctrl Break". After entering this command you will see the symbol "OK" which is a BASIC language prompt. Depress "function button 2" (F2) to rerun the particular program. If additional trouble is encountered, start the entire program over by entering the following commands:

1. ctrl break
2. system (enter)
3. design (enter).

CHAPTER TWO - PRELIMINARY ESTIMATE OF TAKE-OFF WEIGHT

A. DISCUSSION

The "Request for Proposal" provides a mission profile for the aircraft to perform. This profile must be fitted to the prescribed format. The design program uses this format to obtain an estimate of take-off weight. This chapter is a computerized version of Nicolai's Chapter 5. The following phases are available:

Phase 1 - engine start and take-off
Phase 2 - accelerate to cruise velocity and altitude
Phase 3 - cruise out to destination
Phase 4 - accelerate to high speed dash
Phase 5 - combat
Phase 6 - return cruise
Phase 7 - loiter.

T.O.	Climb	Cruise	Accel	Combat	Cruise	Loiter/land
1	2	3	4	5	6	7

Each phase must be completed in order. If a phase is inappropriate it may be effectively deleted by entering zero for the time, distance or acceleration as appropriate. It is assumed that the specified ordnance is dropped during the combat phase.

B. MISSION PROFILE CHART

MISSION PROFILE CHART

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```
                 W3         W4         W5        W6         W7
                 |---------|---------|--------|---------|
                /  Cruise     Accel    Combat  Cruise    \
               /   outbound                     inbound   \
              /                                            \
             / Climb                             loiter     \
|---------|                                                  |-----|
WTO       W2                                                   W8
Take-off                                                       Land
```

1. Cruise outbound distance = ______________ nm.
2. Cruise outbound altitude = ______________ ft.
3. Accelerate to = ______________ mach
4. Combat time = ______________ sec.
5. Cruise inbound distance = ______________ nm.
6. Loiter time = ______________ min.
7. Ordnance loaded = ______________ lbs.
8. Ordnance dropped = ______________ lbs.

C. PRELIMINARY ESTIMATES

Preliminary Estimates

****

Now make a preliminary estimate for the minimum WTO necessary to fly the above profile. Use historical references such as Jane's "All the World's Aircraft" and Appendix B.

Initial guess for WTO = ______________ lbs.

Select an engine from an appropriate reference source and fill in engine data below.

1. Engine designation ______________________
2. Cruise SFC (approx) ______________________
3. Military SFC ______________________
4. Combat (afterburner) SFC ______________
5. Loiter SFC ______________________
6. Engine weight ______________________

D. MISSION REQUIREMENTS CHART

Mission Requirements Chart

This chart summarizes all of the data required to run Chapter Two of the design program. Use the information gathered in sections 1 and 2 and the RFP to complete the following list.

Phase I. Engine Start and Take-off

1. W2/WTO ____________________
2. WTO (preliminary estimate) ____________________
3. Ordnance loaded ____________________
4. Ordnance expended ____________________
5. Reserve fuel fraction ____________________
6. Trapped fuel fraction ____________________
7. Number of crew ____________________
8. Weight per crewman ____________________
9. Composite savings percentage ____________________

Phase II.

10. Mach: Initial cruise ____________________

Phase III. Cruise Outbound

11. Radius outbound (nm.) ________________
12. SFC outbound (lb.fuel/lb.thrust/hr.) ________________
13. Mach outbound (see #10) ________________
14. Initial cruise altitude ________________
15. L/D outbound ________________

Phase IV. Accelerate to High Speed

16. Mach before accel (see #10,13) ________________
17. Mach after acceleration ________________

Phase V. Combat

18. Combat thrust ----------------
19. Combat SFC ----------------
20. Combat seconds ________________

Phase VI. Return Cruise

21. Radius inbound (nm.) ______________
22. SFC inbound ______________
23. Mach inbound ______________
24. Altitude inbound ______________
25. L/D inbound ______________

Phase VII. Loiter/Land

26. Loiter time (minutes) ______________
27. SFC loiter ______________
28. L/D loiter ______________

CHAPTER THREE - MEETING PERFORMANCE REQUIREMENTS

A. DISCUSSION

The next step in the conceptual design process is to meet the various performance requirements, making a determintaion of the required thrust/weight ratio and the best wing loading. Knowing take-off weight, and wing loading, the user is able to make a preliminary engine selection and size the wing.

Five performance areas are addressed by the program:

1. take-off requirments
2. climb requirements
3. cruise requirements
4. maneuvering requirements
5. landing requirements

The results from these five sections allow the user to create a performance matching graph as shown in figure A.2. The input requirements are listed in the following sections.

(Note: To plot the results of these sections on a common graph it is necessary that all wing loadings and thrust/weight ratios refer to a common reference. This reference is usually the take-off wing loading and the take-off thrust/weight ratio. For example, if landing wing loading is 80% of the take-off wing loading, the landing wing loading

must be divided by .8 to be plotted on a performance matching graph which has take-off wing loading as the reference. The design program prompts the user for these normalizing fractions and makes the required adjustments.)

B. TAKE-OFF REQUIREMENTS

1. **Take-offdistance** ____________
2. CLmax (take-off configuration) ____________
3. Density ratio ____________
4. Thrust Fraction (available/reference) __________

C. CLIMB REQUIREMENTS

1. Desired final altitude ____________
2. Time to climb (seconds) ____________
3. CDO ____________
4. Aspect Ratio ____________
5. Wing efficiency factor ____________
6. Thrust fraction (start climb/reference) __________
7. Weight/fraction (start climb/reference) __________

D. CRUISE REQUIREMENTS

1. Thrust fraction (cruise/reference) ________
2. Weight fraction (cruise/reference) ________
3. CDO ________
4. Aspect Ratio ________
5. Wing Efficiency factor ________
6. Altitude ________
7. Mach number during cruise ________

D. MANEUVERING REQUIREMENTS

1. Thrust fraction (maneuvering/reference) ________
2. Weight fraction (maneuvering/reference) ________
3. CDO ________
4. Aspect Ratio ________
5. Wing efficiency factor, (e) ________
6. Altitude ________
7. G-load ________
8. Mach ________

E. LANDING REQUIREMENTS

1. Total landing distance ________
2. Density ratio ________
3. CLmax ________
4. Approach Safety Factor ________
5. Weight fraction (landing/reference) ________

CHAPTER FOUR - ASPECT RATIO OPTIMIZATION

A. DISCUSSION

For purposes of the design program, the selection criterion used for aspect ratio optimization was minimum take-off weight. Three methods are available.

1. North American method
2. Fixed mach method
3. Variable Mach method

B. NORTH AMERICAN METHOD

1. Take-off wing loading ____________
2. Wing efficiency factor ____________
3. CDO outbound ____________
4. CDO inbound ____________
5. Reference Aspect Ratio ____________

C. FIXED MACH METHOD

1. Take-off wing loading ____________
2. Wing efficiency factor ____________
3. CDO outcound ____________
4. CDO inbound ____________
5. Reference aspect ratio ____________

D. VARIABLE MACH METHOD

1. CDO outbound ____________
2. CDO inbound ____________
3. Wing efficiency factor ____________
4. Take-off wing loading ____________
5. SFC at mach 0.5 ____________
6. SFC at mach 0.9 ____________
7. Reference aspect ratio ____________

CHAPTER FIVE - WING GEOMETRY

This chapter solves wing geometry equations. Calculations are available for equations presented in Figures A.2 and A.3.

WING GEOMETERY FORMULAS: Part 1
****

Section 2: Sweep Angle Leading Edge, degrees ($Sweep_{LE}$)

Given: design mach (DM)

Assumption: Supersonic wing with subsonic leading edge. Wing swept five degrees behind the mach line.

Formula: $$Sweep_{LE} = 95 - \tan^{-1}\left(\frac{1}{(DM^2 - 1)}\right)$$

Section 3: Sweep Angle 1/4 chord ($Sweep_{C/4}$)

Given: a. Sweep angle leading edge ($Sweep_{LE}$)
b. Taper ratio (L)
c. Aspect Ratio (AR)

Formula: $$\tan(Sweep_{C/4}) = \tan(Sweep_{LE}) - \frac{(1 + L)}{(AR)\ \ (1 - L)}$$

Section 4: Wing Area

Given: a. Take-off weight (WTO)
b. Take-off wing loading (WSTO)

Assumption: none

Formula: $S = WTO / WSTO$

Figure A.2 Wing Geometry Part 1

WING GEOMETRY FORMULAS: Part 2

****

Section 5: Span (b)

Given: a. Aspect ratio (AR)
b. Wing surface area (S)

Assumption: trapezoidal wing

Formula: $b = \{ (AR)\ (S) \}^{.5}$

Section 6. Root Chord (CR) and Tip Chord (CT)

Given: a. Wing surface area (S)
b. Wing span (b)
c. Taper ratio (L)

Assumption: trapezoidal wing

Formulas:

$$Cr = \frac{(2)\ (S)}{(b)\ (1 + L)}$$

$$Ct = (Cr)\ (L)$$

Section 7: Mean Aerodynamic Chord (MAC)
Spanwise distance to Center of Pressure (Ybar)

Given: a. Wing span (b)
b. Taper ratio (L)
c. Root chord (Cr)

Assumption: trapezoidal wing

Formulas: $$MAC = \left(\frac{(2)\ (Cr)}{(3)}\right) \left(\frac{(1 + L + L^2)}{(1 + L)}\right)$$

$$Ybar = \left(\frac{(b)}{(6)}\right) \left(\frac{[1 + (2)(L)]}{(1 + L)}\right)$$

Figure A.3 Wing Geometry Formulas: Part 2

CHAPTER SIX - FUSELAGE LENGTH

A. DISCUSSION

Fuselage lengths are predicted by using WTO and empirical relationships.

B. FUSELAGE LENGTH FORMULAS

1. Jet Fighter

   Fuselage length = $(0.83)\ (WTO)^{0.39}$

   or

   Fuselage length = $(41.0) + (0.00034)\ (WTO)$
   (supersonic aircraft only)

2. Jet Trainer

   Fuselage length = $(0.79)\ (WTO)^{0.41}$

CHAPTER SEVEN - VERTICAL TAIL DESIGN

A. DISCUSSION

This chapter solves the iterative problem of sizing the vertical tail to meet a specific tail volume coefficient. Note: When computing vertical tail aspect ratio, treat the tail as though a mirror image other half were present, and then use conventional wing aspect ratio formulas. The entry for item #7 (wing surface area) should be the actual surface area for the vertical tail, without the mirror image half.

B. INPUT REQUIRMENTS

1. Desired tail volume coefficient ____________
2. Fuselage length ____________
3. CG position on fuselage (ft aft of nose) ____________
4. Wing sweep ____________
5. Wing aspect ratio ____________
6. Wing taper ratio ____________
7. Wing surface area ____________
8. CG position as a fraction of MAC ____________
9. Distance of tail form end of fuselage ____________
10. Tail sweep ____________
11. Tail taper ratio ____________

CHAPTER EIGHT - DETERMINING STRUCTURAL WEIGHTS

A. DISCUSSION

Chapter Eight solves empirical weight estimation formulas for six structural components. The components are:

1. Wing
2. Horizontal Tail
3. Vertical Tail
4. Fuselage
5. Main landing gear
6. Nose landing gear

The required inputs for these components and the empirical formulae are listed in Sections B-G. Historical values are provided for the fuselage, main landing gear and nose landing gear in Figures A.4, A.5 and A.6 in Section H.

B. WING

$$\text{Wing Weight} = (0.0103)\,(K.DW)\,(K.VS)\,(K.FOLD)\,(W.DG*N.Z)^{.5}\,(S)^{.622}\,(AR)^{.785}\,(T.CR)^{-.4}\,(1 + LAMBDA)^{.050}\,(\cos GAMMA)^{-1.0}\,(S.CS)^{.040}$$

1.	K.DW	(.768 delta wing, 1.0 non-delta wing)	________
2.	K.VS	(1.19 variable sweep, 1.0 fixed wing)	________
3.	K.Fold	(1.1 with fold, 1.0 no fold)	________
4.	W.DG	(design gross weight - lbs) (approximately WE + WF)	________
5.	N.Z	(ultimate laod factor) (typically 10-12)	________
6.	S	(wing area - ft sq)	________
7.	AR	(wing aspect ratio)	________
8.	T.CR	(wing thickness divided by root chord)	________
9.	LAMBDA	(wing taper ratio)	________
10.	GAMMA	(wing sweep at 25% chord)	________
11.	S.CS	(area - wing mounted control surfaces) (typically 20-30% of wing area)	________

C. HORIZONTAL TAIL

$$\text{Horizontal tail weight} = (3.316)\,(1 + F.W/B.H)^{-2.0}\left(\frac{(W.DG * N.Z)}{(1000)}\right)^{.260}(S.HT)^{.806}$$

1. F.W (fuselage width at horizontal tail) ____________
2. B.H (horizontal tail span) ____________
3. W.DG (design gross weight) ____________
4. N.Z (ultimate load factor) ____________
5. S.HT (gross horizontal tail area) ____________

D. VERTICAL TAIL

$$\text{Vertical tail weight} = (.879)\,(K.RHT)\,(1 + H.T/H.V)^{.500}$$
$$(W.DG * N.Z)^{.434}\,(S.VT)^{.560}\,(M)^{.414}\,(L.T)^{-.789}$$
$$(1 + S.R/S.VT)^{.150}\,(AR.VT)^{.232}\,(1 + LAMBDA.VT)^{.250}$$
$$(\cos GAMMA.VT)^{-.333}$$

1. K.RHT (1.2 for differential UHT, 1.0 for others) ______
   (UHT - single piece horizontal tail)
2. H.T (height, horizontal tail above fuselage) ______
3. H.V (height of vertical tail above fuselage) ______
4. W.DG (flight design gross weight) ______
5. N.Z (ultimate load factor) ______
6. S.VT (vertical tail area) ______
7. M (maximum Mach number) ______
8. L.T (tail lenght - ft) ______
9. S.R (rudder area - sq ft) ______
10. AR.VT (vertical tail aspect ratio) ______
11. LAMBDA.VT (vertical tail taper ratio) ______
12. GAMMA.VT (sweep angle of vertical tail 25% chord) ______

E. FUSELAGE

Fuselage weight = (0.3197) (K.DWF) (W.DG * N.Z)$^{.50}$ (L)$^{.50}$ (D)$^{.250}$ (B)$^{.40}$

1. K.DWF (.80 for delta wing aircraft)
   (1.0 for non-dleta wing aircraft) ________
2. W.DG (flight design gross weight ________
3. N.Z (ultimate load factor) ________
4. L (fuselage structural length) ________
5. H (fuselage structural height) ________
6. B (fuselage structural width) ________

F. MAIN LANDING GEAR

$$\text{Main landing gear} = (K.CB)(K.TP)\frac{(W.L^{*}\ V.SNK^{2})^{.250}}{(S.OM)}$$

$$(L.M)^{1.165}$$

1. K.CB (2.250 for cross beam (F-111 type gear)
   (1.0 for others) -------
2. K.TP (.682 tripod type gear, 1.0 for others) ________
3. W.L (Landing design gross weight) ________
4. W.DG (flight design gross weight) ________
5. V.SNK (landing sink speed - ft/sec ________
6. S.OM (oleo stroke - inches) ________
7. L.M (length of main landing gear) ________

G. NOSE LANDING GEAR

$$\text{Nose landing gear} = (K.2P)\ (W.L * N.L)^{.290}\ (L.N)^{.5}\ (N.NW)^{.525}$$

1. W.L (landing gross weight) ______
2. K.2P (1.246 two position nose gear, 1.0 others) ____
3. N.L (ultimate landing load) ______
4. L.N (nose gear lenght - inches) ______
5. N.W (number of nose wheels) ______

H. HISTORICAL VALUES

The following data was obtained from the Vought Weight Estimation Manual. [7:4.5]

FUSELAGE
****

Aircraft	K.DWF	W.DG	N.Z	L	D	B	W.F
1. F-105	1.0	34768	13.0	64.4	6.3	8.3	5780
2. F-106	1.0	30590	9.0	63.2	6.5	8.1	4401
3. F-111	1.0	59000	9.8	58.2	7.1	12.2	10870
4. F-4K	1.0	37500	9.8	46.0	6.3	8.3	5185
5. F-5B	1.0	11087	10.1	44.2	5.0	5.9	2176
6. F-8E	1.0	26000	9.6	53.0	5.9	4.7	3555
7. A-4E	0.8	12504	10.5	39.6	5.0	5.3	1434
8. A-5A	1.0	40953	7.5	69.0	4.7	10.7	7456
9. A-6A	1.0	36526	9.8	44.1	7.1	6.2	4047
10. A-7A	1.0	26203	10.5	44.2	7.2	5.0	2996

Figure A.4 Fuselage Historical Values

MAIN LANDING GEAR

****

Aircraft	K.CB	K.TP	W.L	W.DG	V.SNK	S.OM	L.M
1. F-105D	1	1	33560	34768	9.5	9.0	88.2
2. F-106	1	1	26172	30590	9.0	11.7	68.2
3. F-111B	2.25	1	52400	59000	22.8	11.7	34.3
4. F-4K	1	1	36000	37500	24.0	17.4	63.3
5. F-5B	1	1	12200	11087	10.0	10.2	48.3
6. F-8E	1	.682	22000	26000	19.6	7.3	46.5
7. A-4E	1	1	11556	12504	20.0	14.0	53.4
8. A-5A	1	1	32653	40953	21.0	18.0	60.2
9. A-6A	1	1	33386	36526	20.3	15.0	78.8
10. A-7A	1	.682	24431	26203	25.8	8.0	44.1

Figure A.5 Main Landing Gear Historical Values

NOSE LANDING GEAR

****

Aircraft	W.L	W.DG	K.2P	N.L	L.N	N.W
1. F-105	33560	34768	1	4.0	61.2	1
2. F-106	26172	30590	1	4.5	44.5	2
3. F-111B	52400	59000	1	11.5	66.0	2
4. F-4K	36000	37500	1.246	7.15	71.8	2
5. F-5B	12200	11087	1	3.6	40.0	1
6. F-8E	22000	26000	1	8.25	46.2	1
7. A-4E	11556	12504	1	7.17	65.9	1
8. A-5A	32653	40953	1	7.05	60.5	1
9. A-6A	33386	36525	1	6.2	50.4	2
10. A-7A	24431	26206	1	9.6	37.0	2

Figure A.6 Nose landing Gear Historical Values

CHAPTER NINE - REFINED ESTIMATE OF WTO

A. DISCUSSION

Chapter Nine uses the combined weight of the six major components (WS) from Chapter Eight and payload data from Chapter Two data to make a refined estimate for WTO.

B. REQUIREMENTS

The inputs required to perform these calculations are automatically recovered from the data base created by other chapters.

APPENDIX B - ORDERING INFORMATION

For a copy of this program, send a formatted 5.5 inch diskette in a self addressed mailer to:

Lcdr. M. L. Cramer
VF-143
FPO NEW YORK, N.Y. 09501

To run the diskette upon return, a microsoft BASIC language must also be installed. The program runs without problems using IBM BASICA or GWBASIC. The BASIC language program is not provided because of copyright restrictions.

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7. Vought Aeronautics Division, Report No.2-59320/8R-50475, Weight Estimation Manual, by R. N. Stanton, August 1968.

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7-87

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